JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 39, No. 2, March–April 2002

Spacecraft Design Lifetime

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A general discussion of issues that drive and limit spacecraft design lifetime is presented. The effects of varying the spacecraft lifetime requirement on different subsystems are explored, and typical spacecraft mass and cost profiles are deduced. Quantitative analyses confirm that the design lifetime is a key requirement in sizing various subsystems and significantly affects the spacecraft mass and cost to initial operating capability. The analysis introduces a formally defined economic metric, the cost per operational day, to help guide the specification of the design lifetime requirement. Preliminary results suggest that other factors should also be taken into account in specifying the design lifetime, namely, the loss of value resulting from technology obsolescence as well as the volatility of the market the system is serving in the case of a commercial satellite.

Nomenclature

A	=	area of radiator, m^2
g	=	gravitational field strength, m/s^2
I	=	inherent degradation of manufactured solar arrays, typically 80%
I_s	=	solar intensity at 1 astronomical unit, 1367 W/m^2
I_{sp}	=	specific impulse, s
L_d	=	solar array life degradation, %
M_i	=	mass of subsystem i, kg
N	=	number of batteries
n_b	=	transmission efficiency between batteries and load, typically 90%
P_{BOL}	=	power at beginning of life, W
P_{EOL}	=	power at end of life, W
P_e	=	power requirement during eclipse, W
R_i	=	probability that subsystem i is operational
T_e	=	duration of eclipse, h
T_{life}	=	spacecraft design lifetime, years
$T_{rad\text{-}max}$	=	maximum allowable temperature for the radiator, K
α	=	absorptance of the radiator
ΔV	=	velocity increment, m/s
ε	=	emittance of radiator
η	=	solar cells energy conversion efficiency (19% for GaAs and 15% for Si cells) times array inherent degradation
η_r	=	radiator efficiency, typically 90%
θ	=	solar aspect angle, rad
λ_i	=	failure rate of subsystem i
ρ_i	=	component i density, kg/m^3
σ	=	Stefan–Boltzmann constant, 5.68×10^{-8} W/m^2 · K^4

Introduction

IN recent years, several space programs have chosen to increase their space segment design lifetime. Over the last two decades, telecommunications satellites, for instance, have seen their design lifetime increase on average from 7 to 15 years. This trend is also observed in the design and development of many high-value assets, for example, the average life span of a helicopter delivered today can exceed 30 years or 20,000 h of operation.

In most cases, increasing the space segment design lifetime was driven by the desire to maximize the return on investment (ROI). However, extending the design lifetime has several side effects. First, doing so leads to larger and heavier satellites as a result of several factors, such as additional propellant for orbit and station keeping, power generation, and storage. This additional mass in turn increases the satellite's development and production costs. Second, as the satellite design lifetime increases, the likelihood that the satellite becomes obsolete, technically and commercially, before the end of its mission increases. In many cases, the initial circumstances from which the original system requirements were derived change or are modified during the spacecraft's operational lifetime. Setting a spacecraft design lifetime requirement can, therefore, be a critical task for system designers. However, what drives spacecraft design lifetime? How do designers, managers, and or customers decide on spacecraft design lifetime? Are there any economical considerations such as minimizing cost per operational day or maximizing the ROI for establishing this requirement, or is the design lifetime mainly dictated by technical limitations? Some of the technical considerations that limit spacecraft lifetime include the depletion of consumables, degradation due to spacecraft/environment interaction (of solar panels, radiators, electronics, etc.), wear and tear (thrusters pulse life, wheel bearings, etc.), and reliability/redundancy issues.

Consider, for instance, AT&T's Telstar 3 communications satellites based on the Hughes Space and Communications Co. HS-376 bus. These satellites have 10-year design lives, a significant increase over the 7-year lives for earlier satellite models. Life extension was made possible by the use of improved nickel–cadmium batteries and the introduction of solid-state power amplifiers in place of traveling wave tubes.

Questions regarding the design lifetime requirement can fall into the three following categories: 1) What limits the design lifetime? 2) How does the total system mass and cost change as a function of the design lifetime requirement? 3) What does the customer ask the contractor to provide for a spacecraft design lifetime and why?

Although related, these three questions nevertheless cover different realities. The first question addresses the issue of the lifetime boundary: How far can designers push a spacecraft's design lifetime and why can it not be extended any further? The technical considerations listed earlier dictate this boundary.

The second question, closely related to the first one, focuses on the effects of varying the design lifetime requirement on the total spacecraft mass and cost, or the cost to initial operating capability (IOC). It is clear that the design lifetime will have a strong bearing on the mass and cost of the system by affecting the power budget [beginning of life (BOL) solar panel, battery capacity, etc.], the propellant budget (orbit and station keeping), the level of redundancy, and other key system parameters.

The third question builds on the two preceding ones: Given the maximum achievable design lifetime, as well as the impact of the mission duration on the spacecraft mass and cost, what does the

Received 16 May 2001; revision received 30 August 2001; accepted for publication 12 September 2001.

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customer require for spacecraft design lifetime? The design lifetime should not necessarily be set to the maximum achievable value. For example, a commercial customer may not want to make the contract life of a spacecraft too long. New or enhanced payload capabilities, for example, a better spatial resolution for an optical instrument, might be developed and become available within a couple of years following deployment, hence, the need to launch a new satellite or risk losing market share to a competitor who launches later with newer and more advanced capabilities.

The answer to the first question can be easily obtained by examining the maximum operational lifetimes of the various technologies embedded in the spacecraft. In this paper, we investigate question 2, that is, how does the spacecraft lifetime requirement impact the design and sizing of the various subsystems? How do these subsystems scale with the design lifetime, and consequently, how does the total spacecraft mass and cost scale with the design lifetime requirement? Answering this question is pivotal for understanding the rationale in specifying the lifetime requirement; hence, it is a prerequisite for answering question 3.

This paper is organized as follows. First, typical percentage mass contributions of different subsystems to the total spacecraft mass are presented, and the main mass drivers are identified. Second, the impact of the design lifetime requirement on the sizing of the spacecraft subsystems is investigated. The results are then integrated, and typical spacecraft mass profile as a function of the design lifetime are presented. These mass profiles are in turn transformed into cost profiles, and the cost per operational day metric is introduced. Finally, the limitations of the analysis are addressed, and the implications of this analysis are discussed.

Typical Mass Distribution of Satellites

To assess the impact of design lifetime on spacecraft mass, it is useful to identify the mass contributions of the subsystems to the spacecraft total mass. Table 1 shows some historical spacecraft mass distribution data. For example, the electrical power subsystem (EPS) accounts on average for 30% of satellite dry mass, with a standard deviation of 7%. The EPS, along with the payload and spacecraft structure, are the major mass contributors and make up approximately 80% of satellite dry mass.

Spacecraft Subsystems and Design Lifetime

We now examine how different subsystems scale with the design lifetime requirement. This requirement is a key parameter in sizing several spacecraft subsystems: It directly impacts the design and sizing of some subsystems, for example, the EPS, and indirectly impinges on others, for example, the structure. These influences and couplings are qualitatively captured in Table 2. The diagonal in Table 2 represents the direct impact of the design lifetime requirement on each subsystem. The off-diagonal terms read as follows. Subsystems in the first column scale with the design lifetime, driven by changes in subsystems in the first row. The number of crosses represents the degree of influence (+ + + major influence, + minor influence).

EPS

The EPS generates power, conditions and regulates it, stores it for peak demand or eclipse operation, and distributes it throughout the spacecraft.[1] The design lifetime is a key parameter in sizing the EPS. It directly impacts 1) the life degradation of the solar arrays, hence, their surface, and, consequently, their mass and 2) the battery capacity through the extended number of cycles, hence, reduced depth of discharge (DOD) with design life. The design lifetime also indirectly impacts the sizing of the power controllers and regulators as well as the harnesses and cabling that interconnect the spacecraft subsystems. In this subsection, we explore how the EPS scales with the design lifetime.

Solar Arrays

In designing solar arrays, experts typically trade mass, surface, and cost. "Life degradation L_d of solar arrays occurs because of thermal cycling in and out of eclipses, micrometeoroid strikes, plume impingement from thrusters, material outgassing, and radiation damage throughout the duration of the mission."[1] Life degradation is a function of the design lifetime and can be estimated as follows:

$$L_d = (1 - \text{degradation/year})^{T_{\text{life}}} \tag{1}$$

The degradation per year is a function of the spacecraft orbital parameters (location with respect to the Van Allen belts), as well as the solar cycle. Typically, for a silicon solar array in low Earth orbit (LEO), power production can decrease by as much as 3.75% per year and for gallium–arsenide 2.75% (Ref. 1).

Figure 1 shows typical life degradations of silicon solar arrays and gallium–arsenide arrays as a function of design lifetime. Gallium–arsenide cells are both more efficient (19% energy conversion efficiency) and degrade slower than silicon cells (efficiency of about 15%). For instance, given a six-year design lifetime, the power output of silicon arrays will degrade by 80% and for gallium–arsenide arrays by 85%. The array's performance at the end of life (EOL) is given by

$$P_{\text{EOL}} = P_{\text{BOL}} \times L_d \tag{2}$$

Table 1 Percentage mass distribution, averages and standard deviations (adapted from Ref. 1)

	Percentage of satellite dry mass (standard deviation)						
System	EPS	Payload	Structure	ADCS	TT&C	Propulsion	Thermal
Communication	32 (5)	27 (4)	21 (3)	7 (2)	5 (2)	4 (1)	4 (2)
Navigation	32 (3)	21 (2)	23 (3)	6 (0.5)	5 (1)	3 (0.5)	10 (1)
Remote sensing	25 (4)	36 (5)	20 (3)	5 (2)	4 (1)	7 (3)	3 (1)
Average	30 (7)	28 (7)	21 (5)	6 (3)	5 (2)	4 (3)	6 (2)

Table 2 Design lifetime influence matrix

Subsystem	ADCS	TT&C	EPS	Thermal	Structure	Propulsion	Propellant
ADCS			+ + +	+	+ + +	+	+ +
TT&C		Redundancy, shielding					
EPS			Solar array degradation, batteries' DOD				
Thermal			+ + + ($\Delta P_{\text{BOL-EOL}}$)	Degradation of thermal properties of coating			
Structure	+	+	+ + +	+		+	+ +
Propulsion						Wear-and-tear/ on–off cycles	
Propellant	+	+	+ + +	+	+ + +	+	Increase in ΔV with design lifetime

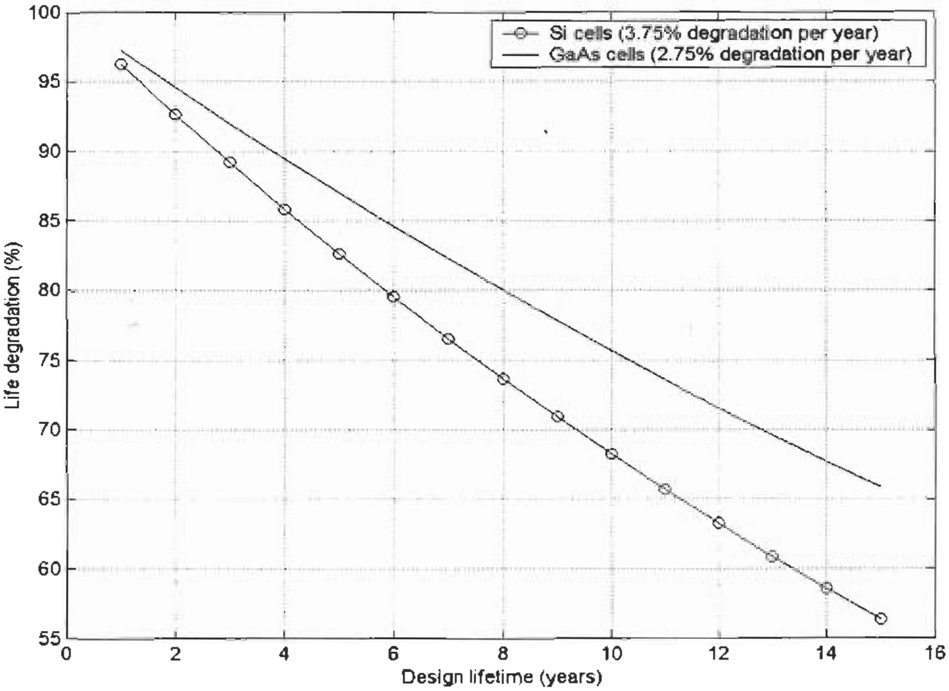

Fig. 1 Typical life degradation of solar arrays in LEO as a function of design lifetime.

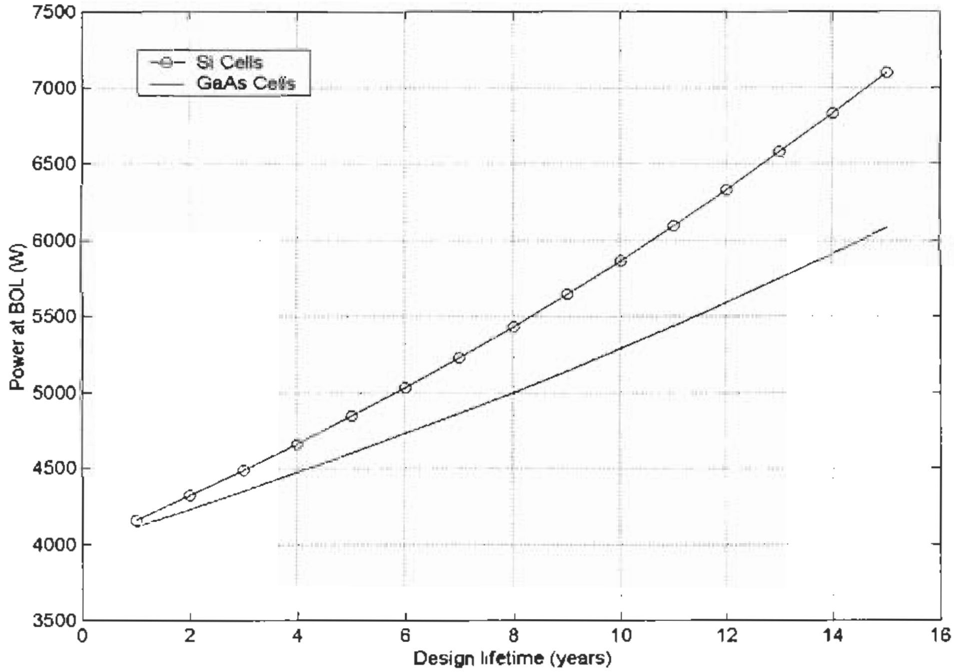

Fig. 2 Solar array P_{BOL} as a function of design lifetime for a 4-kW P_{EOL} requirement.

Given a power requirement at EOL, the power output of the solar arrays at BOL scales inversely with life degradation L_d, and the solar arrays have to be overdesigned to accommodate this performance degradation. Figure 2 shows the relationships between the P_{BOL} and the design lifetime for a 4-kW P_{EOL} requirement. For example, to deliver 4 kW at the EOL of a 10-year mission, the solar arrays should be designed to provide approximately 5.5 kW in the case of GaAs cells and 6 kW in the case of Si cells at the BOL.

The solar array area required to produce the P_{BOL} is approximately

$$S_{sa} \cong P_{BOL}/(I_s \times \eta) \tag{3}$$

Given the specific performance of the array in watt per kilogram (or watt per square meter), the mass of a planar array can be directly evaluated. Typical specific performances range between 20 and 70 W/kg. Results for a nominal specific performance of 40 W/kg are presented in Fig. 3. For instance, to deliver 4 kW at the EOL of a 10-year mission, the solar arrays would weigh approximately 130 kg, that is, 22 kg in excess of a solar array delivering the same 4 kW at the EOL of a 3-year mission. This is equivalent to approximately 20% mass penalty for seven extra years of life.

Batteries

Spacecraft in Earth orbit undergo between 90 and 5500 eclipses per year. The former figure is typical of a geostationary Earth orbit (GEO) satellite, the latter for a satellite in LEO. During eclipse, electric power is supplied by secondary batteries that are recharged by the solar arrays when the spacecraft reemerges into sunlight. In addition, there are some instances when batteries are called upon to

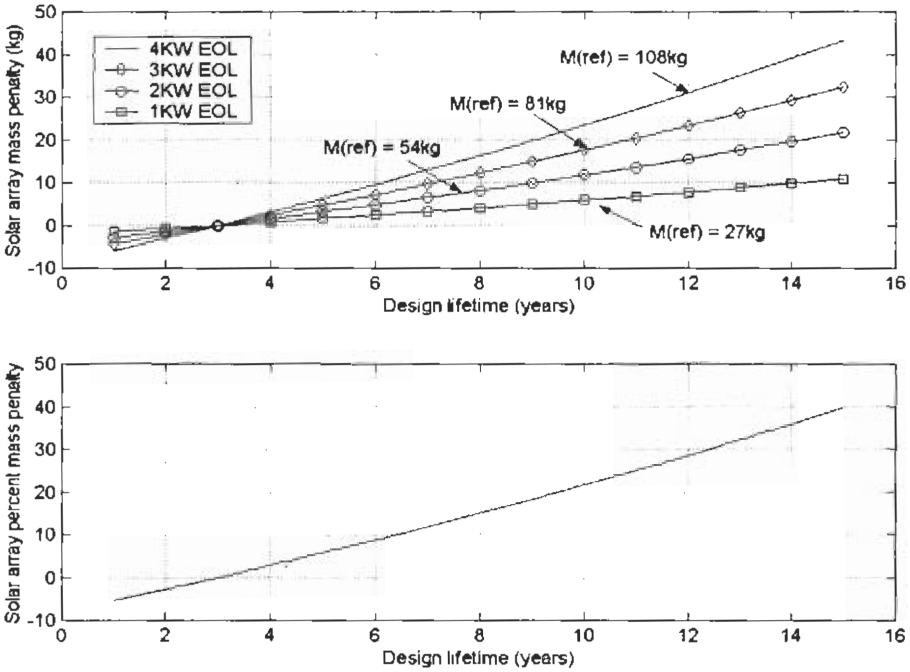

Fig. 3 Solar array (GaAs) mass, mass penalty, and percent mass penalty as a function of the design lifetime; reference mission is three years.

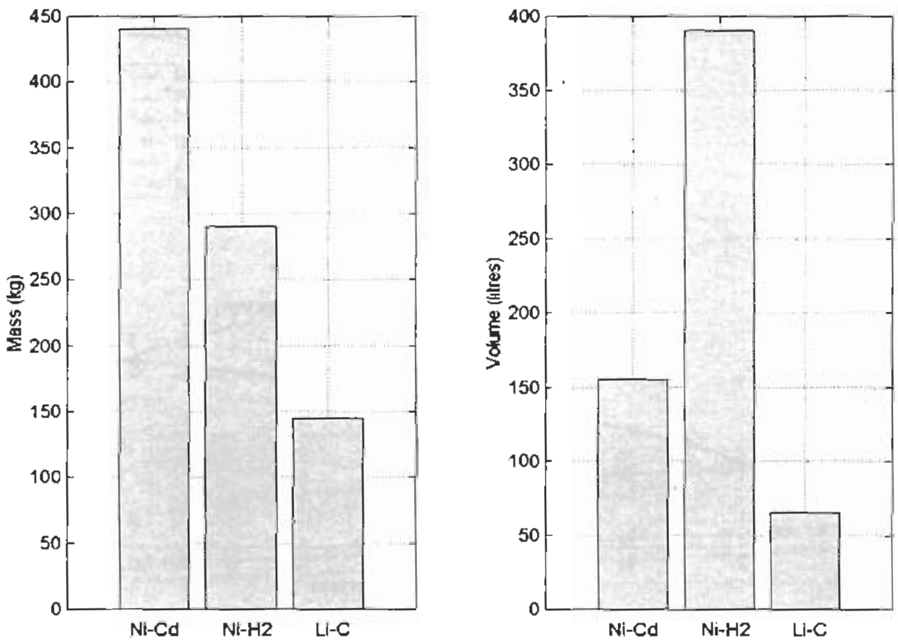

Fig. 4 Mass and volume of different types of batteries for a 10-kWh capacity.[2]

provide peak power in sunlight periods. The existing state-of-the-art and space-qualified batteries (nickel–hydrogen) are heavy and can constitute up to 15% of the dry mass of a typical communications satellite.[2] Current secondary battery technology includes nickel–cadmium, which is a very common space-qualified secondary energy storage system. Nickel–hydrogen batteries are currently the energy storage system of choice for most aerospace applications where high specific energies and long design life are required. Lithium–ion and lithium–carbon batteries are currently under development with expected space qualification for GEO and LEO applications by 2005–2010.[1] Lithium–ion and lithium–carbon batteries would offer significant mass and volume reduction compared to nickel–cadmium and nickel–hydrogen technology, as shown in Fig. 4.

The design lifetime significantly impacts the sizing of secondary batteries. Indeed, the amount of energy available from secondary batteries, the DOD, decreases with the number of cycles of charging and discharging. To first order, the number of charge/discharge cycles is equal to the number of eclipses a satellite encounters during its design lifetime. Typically a satellite in GEO undergoes two periods of 45 days per year with eclipses lasting no more than 72 min, hence 90 cycles of charging and discharging per year. Satellites in LEO undergo approximately one eclipse per orbit. For a 90-min orbit, this amounts to 16 eclipses per day, or approximately 5500 cycles per year, with a maximum shadowing period of nearly 36 min per orbit. Figure 5 shows the DOD as a function of the number of charge/discharge cycles a battery undergoes, as well as the DOD as a function of the design lifetime of a satellite in GEO.

For instance, for a 3-year mission in GEO, the average DOD for a nickel–cadmium battery is approximately 76%, but it drops to 62% for an extended mission of 10 years. How does this impact the sizing of the battery? Battery capacity is estimated as follows:

$$C_r = \frac{P_e \times T_e}{(\mathrm{DOD}) \times N \times n} \tag{4}$$

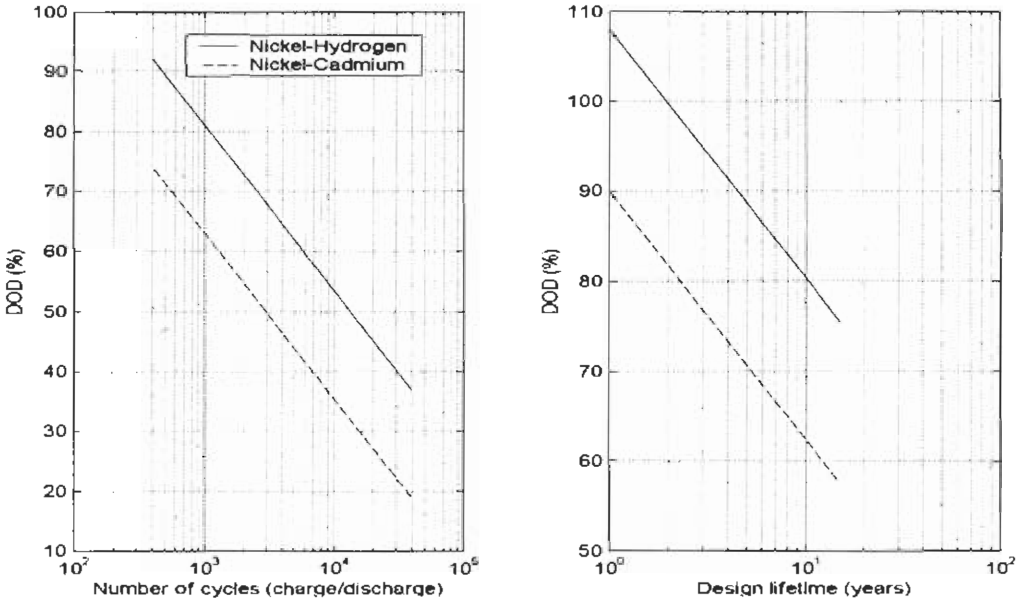

Fig. 5 DOD as a function of number of cycles and design lifetime for a GEO satellite.

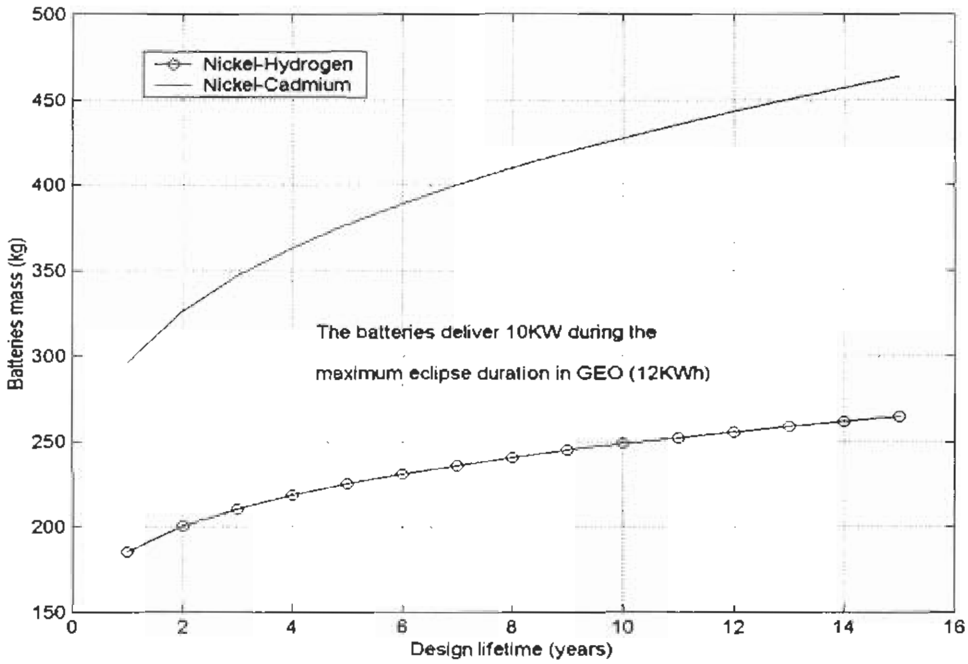

Fig. 6 Mass of 12-kWh battery as a function of design lifetime.

The battery capacity scales inversely with the DOD. Therefore, as the number of cycles or the design lifetime increases, the energy available from the batteries during each cycle decreases, that is, the DOD decreases. Consequently, the batteries have to be overdesigned as the design lifetime increases. The mass of batteries can be obtained given the specific energy density of the battery. For nickel–cadmium batteries, the specific energy density ranges between 25 and 30 kWh/kg and for nickel–hydrogen between 40 and 60 kWh/kg (Fig. 6). Lithium–ion and lithium–carbon are expected to reach the 100-kWh/kg level. Figure 7 shows the evolution of battery mass as a function of design lifetime. The power delivered during eclipse is kept constant.

Figure 6 shows the advantage of nickel–hydrogen batteries over nickel–cadmium for high-energy capacity requirements and long mission duration. For smaller capacities, the mass of the nickel–hydrogen batteries, the mass penalty, and the percent mass penalty considering a three-year reference mission as a function of the design lifetime are given in Fig. 7.

Power Control Unit, Cables, and Harnesses

The power distribution system (or subsubsystem) consists of cabling, fault protection, and switches in the form of mechanical or solid-state relays to turn power on and off to the spacecraft loads. Power regulation is required for two main tasks: 1) controlling the solar array power output to prevent battery overcharging and spacecraft heating and 2) regulating the spacecraft power bus voltage (or each load separately).

The solar array output is described by a plot of current (i) vs voltage (v). This i–v curve changes both due to seasonal variation in the array temperature and the solar intensity and due to radiation degradation of the solar cells as already discussed. The array voltage is maximum as the spacecraft comes out of eclipse when the temperature of the cells is minimum; hence the need to regulate the solar array output.[3] An unregulated bus has a voltage that varies significantly. This is unacceptable for most of the electronic equipment of the payload and the spacecraft if voltage regulation is not provided separately at each load or equipment. Voltage regulators

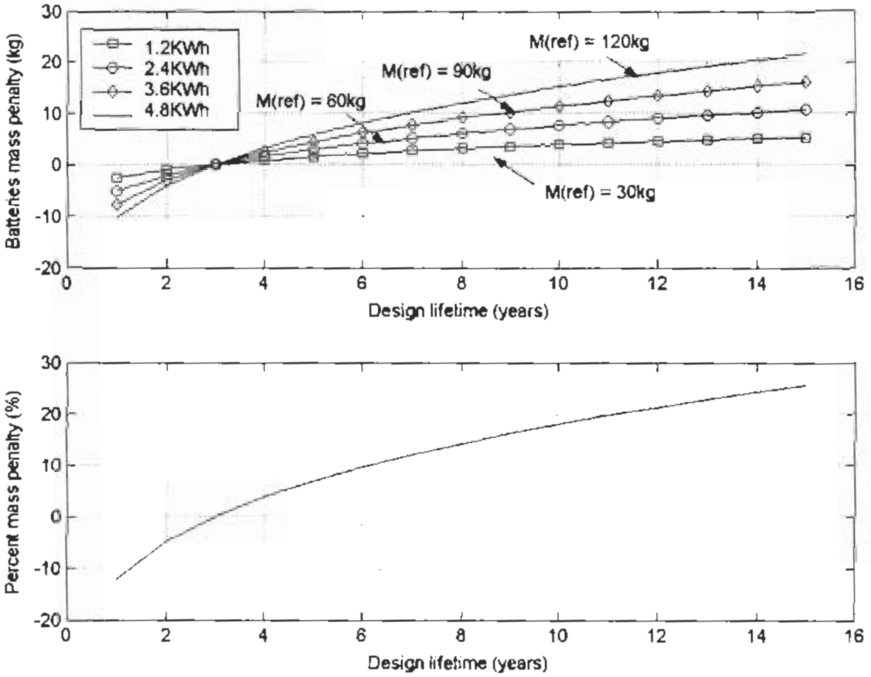

Fig. 7 Nickel–hydrogen batteries, mass, mass penalty, and percent mass penalty as a function of the design lifetime; reference mission is three years.

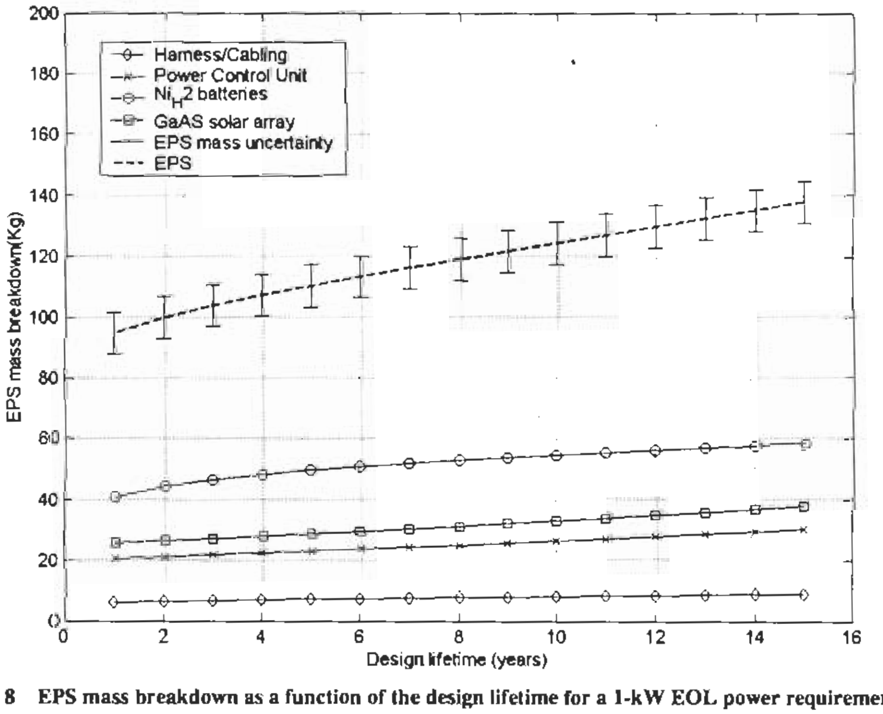

Fig. 8 EPS mass breakdown as a function of the design lifetime for a 1-kW EOL power requirement.

and converters are, therefore, placed either separately at each load or on the spacecraft power bus.[1]

It is difficult to quantify how the mass of the power control unit and the power distribution system scale with the design lifetime. The power control unit as well as the cabling and harness are indirectly affected by the design lifetime because excess power is required at BOL. We use a mass estimate relationship to evaluate the mass (kilogram) of the power control unit (PCU) and the power distribution system (watt):

$$M_{\mathrm{PCU}} = 0.0045 \times P_{\mathrm{BOL}} \tag{5}$$

The mass of the power distribution system constitutes a large part of the EPS mass, roughly 10–20%:

$$M_{\mathrm{dist}} = 0.15 \times M_{\mathrm{EPS}} \tag{6}$$

Figure 8 shows a typical mass breakdown of the EPS for a spacecraft in GEO in terms of its components, solar array, batteries, PCU, and power distribution, as a function of the design lifetime:

$$M_{\mathrm{EPS}} = M_{\mathrm{array}} + M_{\mathrm{batteries}} + M_{\mathrm{PCU}} + M_{\mathrm{dist}} \tag{7}$$

The mass, mass penalty, and percent mass penalty for the electrical power subsystem as a function of the design lifetime are given in Fig. 9. The design lifetime for the reference mission is three years.

Caveat

The preceding sections presented a simple design process for sizing the solar arrays and the batteries. A limited number of parameters were considered, as well as two mass estimate relationships, to derive typical mass profiles of the EPS as a function of the design lifetime. These parameters included the power at EOL requirement, the spacecraft orbital parameters (to derive the eclipse duration for the sizing of the batteries and the solar arrays degradation per year), the solar cell type or the cell energy conversion efficiency, the array specific performance (watt per square meter or watt per kilogram), and the battery type or its specific energy density. The purpose of this analysis was to highlight and capture the impact of the design lifetime on the sizing of the EPS in a semiquantitative way. In reality, the design process of the solar arrays and the batteries is much more involved: Designers have a plethora of variables to trade and optimize. More elaborate design processes of the EPS are available in the literature.[1,3]

Thermal Subsystem

A spacecraft contains many components that will function properly only if they are maintained within specified temperature ranges. The thermal design of a spacecraft involves identifying the sources of heat, designing proper heat transfer between all spacecraft elements, and rejecting heat so that different components stay within their operating temperature ranges.[4]

As in the preceding sections, we are interested in how the thermal subsystem's mass scales with the design lifetime. It is useful to keep in mind in the following discussion that the thermal subsystem accounts on average for only 6% of a spacecraft's dry mass (see Table 1). A spacecraft's thermal design is highly dependent on the mission class and the attitude stabilization type. Assuming a configuration of the thermal subsystem has been selected for a reference mission (selection of a passive vs active thermal control, thermal coating and multilayer insulation, heat pipes, louvers, radiators, electrical heaters, etc.), should the subsystem be redesigned if the spacecraft design lifetime varies? If so, how does its mass scale with the design lifetime?

To answer the preceding questions, we first need to look into the different sources of heat that affect a spacecraft. These include solar radiation, Earth albedo and infrared radiation, and equipment power dissipation (electrical components and wiring). Although the first two are not affected by the design lifetime, it was shown earlier that the power requirement at BOL increases as the design lifetime increases due to solar array degradation. This excess power (see Fig. 2) must be handled by the thermal subsystem. Therefore it is reasonable to assume that the thermal subsystem additional mass varies as a function of the difference between P_{BOL} and P_{EOL}:

$$\Delta M_{\mathrm{thermal}} = f(P_{\mathrm{BOL}} - P_{\mathrm{EOL}}, T_{\mathrm{life}}, \ldots) \tag{8}$$

Radiators are sized for the hottest conditions. The heat-balance equation can be written as follows:

$$\varepsilon\sigma T^4_{\mathrm{rad\text{-}max}} \eta_r A = \alpha A I_s \sin(\theta) + P \tag{9}$$

The area of the radiator, and consequently its mass, is proportional to the power dissipation:

$$A = P \Big/ \big[\varepsilon\sigma T^4 \eta - \alpha I_s \sin(\theta)\big] \tag{10}$$

Another effect has to be considered in sizing the radiator surface: the degradation of the thermal properties of its surface. Typically for an optical solar reflector (OSR) covering the radiator panel of a spacecraft in GEO, the solar absorptance and emittance vary as shown in Table 3.

The radiator area has to be sized for the worst case. When the power dissipation is assumed to be a fraction of the electric power delivered by the solar panels, the radiator's surface can be estimated as follows:

$$A = \max\left[\frac{k \times P_{\mathrm{BOL}}}{\varepsilon_{\mathrm{BOL}}\sigma T^4 \eta - \alpha_{\mathrm{BOL}} I_s \sin(\theta)}; \frac{k \times P_{\mathrm{EOL}}}{\varepsilon_{\mathrm{EOL}}\sigma T^4 \eta - \alpha_{\mathrm{EOL}} I_s \sin(\theta)}\right] \tag{11}$$

It is not clear which term dominates in this relationship. The variations of the solar absorptance $\alpha(t)$ and emittance $\varepsilon(t)$ as a function of time depend on several parameters such as the surface material and the orbital parameters of the spacecraft. In the preceding example, the first term drives the sizing of the radiator's area. To first order, we will consider the mass of the radiator to be proportional to P_{BOL}:

$$M_{\mathrm{rad}} = k_1 \times P_{\mathrm{BOL}} \tag{12}$$

Table 3 Thermal properties of OSR at BOL and EOL after seven years in GEO (adapted from Ref. 3)

Period	Solar absorptance α	Solar emittance ε
BOL	0.08	0.85
EOL	0.21	0.85

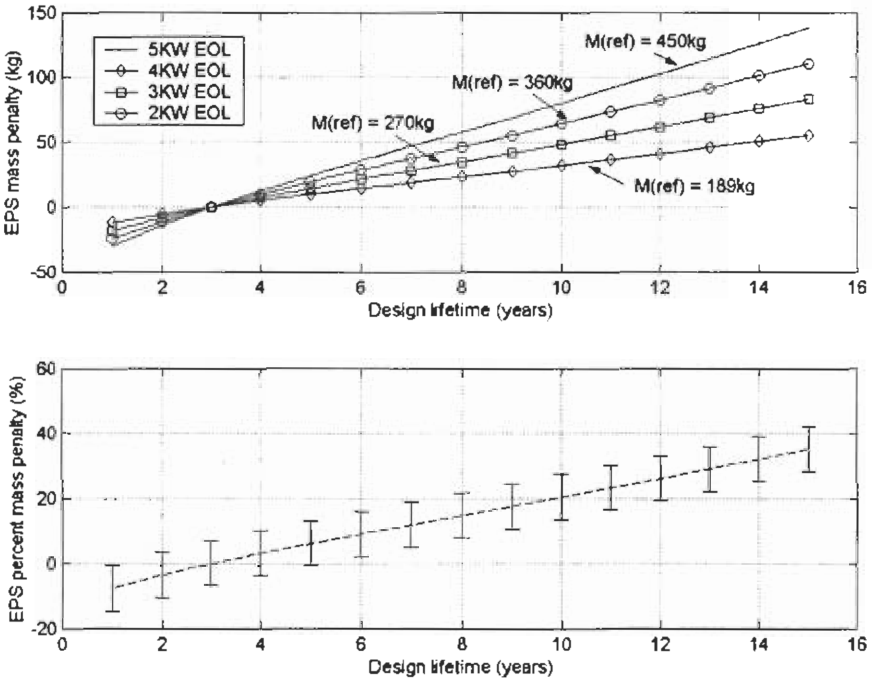

Fig. 9 EPS mass, mass penalty, and percent mass penalty as a function of design lifetime: nickel–hydrogen batteries and GaAs cells, reference mission three years, satellite in GEO.

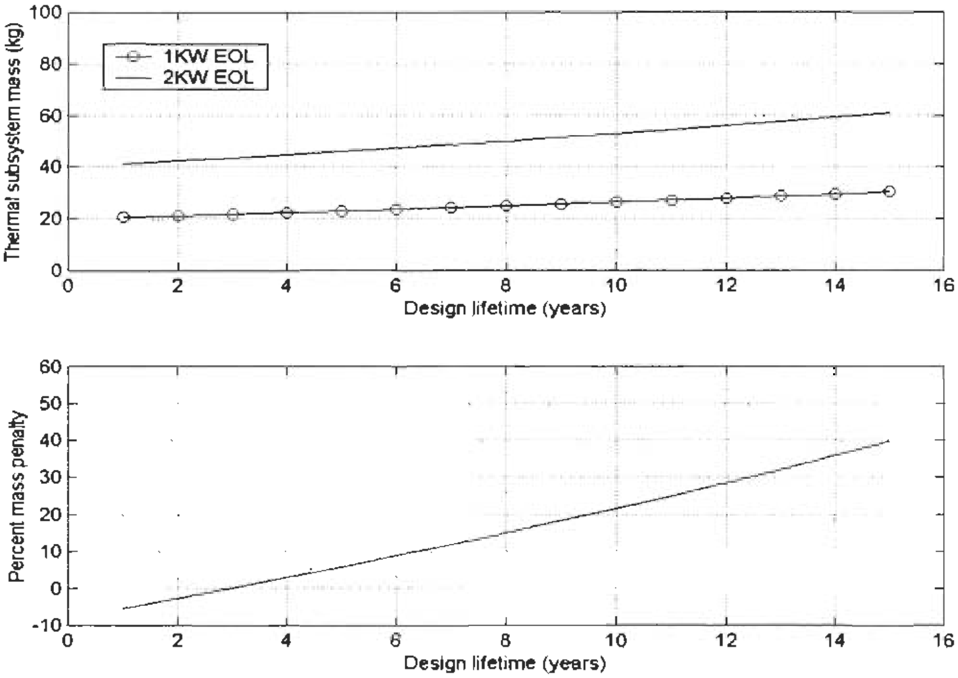

Fig. 10 Thermal subsystem mass and mass penalty as a function of design lifetime.

We will also assume that the rest of the thermal subsystem mass scales with the mass of the radiator:

$$M_{\text{thermal}} = k_2 \times P_{\text{BOL}}$$

For an active thermal control subsystem, values of k_2 that reflect realistic thermal control subsystem mass range between 0.020 and 0.035 kg/W. See Fig. 10.

The preceding discussion represents a first attempt at quantifying the effects of the design lifetime on the thermal subsystem. Although it is clear that the thermal subsystem scales with the design lifetime due to the excess power at BOL and the degradation of the thermal insulation optical properties, nevertheless, it is difficult to quantify those effects in a reasonably accurate way without taking into account a multitude of parameters regarding the spacecraft configuration, the type of thermal control, etc., as well as particular details about the mission. Such details are beyond the scope of this study.

Telemetry, Tracking, and Control Subsystem

The telemetry, tracking, and control (TT&C) subsystem interfaces between the spacecraft and the ground segment. This subsystem provides the hardware required for the reception, processing, storing, multiplexing, and transmission of satellite telemetry data.[5] The command and data handling subsystem (C&DH), often subsumed under the TT&C subsystem, performs two categories of function: It receives, validates, decodes, and distributes commands to other spacecraft subsystems and gathers, processes, and formats spacecraft housekeeping data for downlink or use by the onboard computer.[6]

As in the preceding sections, we are interested in how the TT&C and the CD&H subsystems' mass scale with the design lifetime. Table 1 shows that that those two subsystems account on average for 5% of a satellite's dry mass. As with the thermal subsystem, these are minor contributors to the spacecraft mass. The TT&C design is driven by the following requirements: data rates for command and telemetry, data volume and storage type, uplink and downlink frequencies, bandwidths, receive and transmit power, beamwidth, and antenna characteristics. Selection criteria for TT&C include performance (bit error rate, noise figure, etc.), compatibility with other existing systems, for example, the Tracking and Data Relay Satellite System, as well as technology risk.[6]

These requirements, as well as the selection criteria, do not depend on the spacecraft design lifetime. Therefore, it is reasonable to assume that, to a first order, the mass of the TT&C does not depend on the design lifetime requirement. The same is true for the C&DH subsystem.

This argument breaks down, however, if we consider the effect of radiation on the onboard electronics and the need to provide additional shielding as the design lifetime increases. Furthermore, the reliability required of the C&DH will affect the subsystem mass as the design lifetime increases: Redundant components will be needed to maintain the same level of reliability for an extended lifetime, hence, increasing the amount of hardware and, consequently, the mass of the subsystem.

We will consider that the mass of the C&DH as well as the TT&C subsystems scale with the level of redundancy n. This approach is further elaborated in the following section.

Reliability/Redundancy Issues

This section is, to a large extent, based on Refs. 7 and 8.

The question we seek to answer or gain insight into is as follows: How does the spacecraft mass scale with design lifetime and mission reliability? Design lifetime is the intended operational time of the spacecraft on-orbit. Mission reliability is defined as the probability that the space system will function without a failure that impairs the mission, over a specified period of time or amount of usage. The elementary expression for the reliability of a single product is

$$R = e^{-\lambda t} \tag{13}$$

For a spacecraft composed of n nonredundant elements all equally essential to the spacecraft operation, the overall series reliability is

$$R_s = \prod_1^n R_i = \exp\left(-\sum_i \lambda_i t\right) \tag{14}$$

For n parallel or redundant elements, the overall parallel reliability is

$$R_p = 1 - \prod_1^n (1 - R_i) \tag{15}$$

When the reliability of the elements is the same, Eq. (15) simplifies to

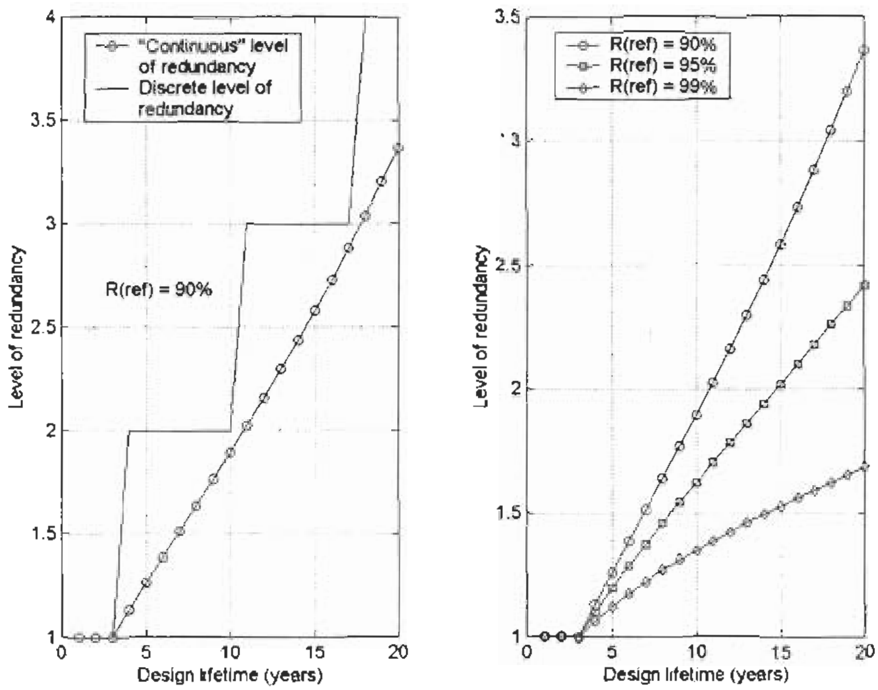

Fig. 11 Level of redundancy as a function of the design lifetime; reference mission is three years (mission reliability or subsystem reliability).

$$R'_p = 1 - (1 - R)^n \tag{16}$$

Consider a reference design lifetime T_{ref} along with a reference mission reliability R_{ref}. As the design lifetime increases, the mission reliability decreases. To maintain the same reliability R_{ref} for an extended duration $T_{life} > T_{ref}$, n redundant elements should be considered:

$$n = \frac{\log(1 - R_{ref})}{\log\left(1 - R_{ref}^{T_{life}/T_{ref}}\right)} \tag{17}$$

This level of redundancy can be calculated per spacecraft subsystem (assuming same components are selected). Consequently, a spacecraft subsystem's mass will scale with its level of redundancy, assuming the customer/designers want to maintain a reliability at EOL, R_{EOL}, that is constant and independent of design lifetime (Fig. 11).

As a simple model, we will assume that the mass of the C&DH/TT&C subsystems scale directly with n.

Propellant Budget

As in the preceding sections, we are interested in how the propellant mass scales with the satellite design lifetime. Propellant is required for orbit change, orbit maintenance, and attitude control. The propellant budget includes propellant to change spacecraft orbital parameters, for example, orbit transfer; to correct for errors due to dispersion injection; to control the attitude during thrusting; to counter disturbance forces, for example, drag in LEO or third-body gravitational attraction in GEO; and to correct spacecraft angular momentum. It also includes a provision for EOL (ΔV to deorbit if the spacecraft is in LEO, or to raise the altitude if the spacecraft is in GEO), as well as a propellant margin that consists of a percentage of the identified propellant requirement.

The total velocity change ΔV_{tot} is converted to propellant mass as follows:

$$M_p = M_0 \lfloor 1 - \exp[-(\Delta V / I_{sp} g)] \rfloor \tag{18}$$

where M_p is the mass of propellant required for a given velocity increment and M_0 is the initial spacecraft mass. As already stated, the satellite ΔV_{tot} is made up of four parts: ΔV_{init} for initial orbit insertion, ΔV_{yr} for yearly station keeping as well as reaction wheel desaturation or unloading, ΔV_{EOL} for EOL disposal, and a ΔV_{margin} (Ref. 6):

$$\Delta V_{tot} = \Delta V_{init} + \Delta V_{yr} \times T_{life} + \Delta V_{EOL} + \Delta V_{margin} \tag{19}$$

We will focus on the first two components of the propellant budget. Two parameters that vary with the design lifetime affect the propellant budget: the satellite initial mass M_0 and the ΔV_{stk} required for station keeping over the mission duration.

For instance, given a ΔV_{init} requirement for orbit transfer from geostationary transfer orbit to GEO, the propellant mass needed to provide this velocity increment is a linear function of M_0, as illustrated in Eq. (18). Because M_0 varies as a function of the design lifetime T_{life} (years), so does $M_{p_{init}}$:

$$M_{p_{init}}(T_{life}) = \psi(\Delta V_{init}, I_{sp}) \times M_0(T_{life}, \ldots) \tag{20}$$

where ψ is given in Eq. (18). For example, for $\Delta V_{init} = 1500$ m/s and an $I_{sp} = 300$ s, $\psi = 0.4$. In other words, the propellant required to perform the orbit transfer accounts for 40% of the spacecraft mass.

The ΔV_{stk} required for station keeping can be estimated as follows:

$$\Delta V_{stk} = T_{life} \times \Delta V_{yr} \tag{21}$$

The ΔV_{yr} yearly for station keeping is a function of the orbit altitude, the solar cycle (minimum or maximum), which in turn alters the atmospheric density, hence the drag encountered by satellite in LEO, or the longitude of station keeping for a satellite in GEO. Typically for a satellite in GEO, $\Delta V_{yr} \sim 50$ m/s. Finally, the propellant mass required to provide ΔV_{tot} is given by

$$M_{p_{tot}} = M_0 \times \left[1 - \exp\left[-\frac{(\Delta V_{init} + \Delta V_{yr} \times T_{life} + \Delta V_{EOL} + \Delta V_{margin})}{g \times I_{sp}}\right]\right] \tag{22}$$

Station keeping is performed using a separate propulsion system from the orbit insertion system. Increasingly ion propulsion or Hall effect thrusters are used for station keeping because of their high specific impulse ($I_{sp} \sim 1500$–3000 m/s). In this case, for a spacecraft in GEO, the mass of propellant required for station keeping per year accounts for 0.2–0.4% of the spacecraft mass at BOL, as opposed to 1.5–3% using the more traditional chemical propulsion system.

Propulsion

The propulsion module subsystem consists of the tanks to hold the propellant, the pipes and pressure-regulating equipment, and the thrusters.[9] As in the preceding sections, we are interested in how the propulsion subsystem's mass scales with the design lifetime, keeping in mind that the propulsion subsystem accounts on average for 4% of a spacecraft dry mass (see Table 1):

$$M_{\text{propulsion}} = M_{\text{tank}} + M_{\text{pipes/valves}} + M_{\text{thrusters}} \tag{23}$$

As the design lifetime increases, the propellant budget increases. Consequently, the volume and mass of the tank necessary to hold the propellant increase. It is reasonable to assume that the other contributors to the propulsion subsystem mass remain unaffected by an increase in the design lifetime.

When a thin spherical tank of thickness e and radius r is assumed, the mass of the tank and the mass of its propellant are

$$M_{\text{tank}} \cong \rho_{\text{tank}}(4\pi r^2) \times e, \qquad M_{\text{propellant}} = \rho_{\text{propellant}}\left(\tfrac{4}{3}\pi r^3\right) \tag{24}$$

Consequently,

$$M_{\text{tank}} = (4\pi\rho_{\text{tank}}e) \times (3/4\pi\rho_{\text{propellant}})^{\frac{2}{3}} \times (M_{\text{propellant}})^{\frac{2}{3}} \tag{25}$$

Finally, we can relate the propulsion subsystem's mass to the propellant mass, which varies as a function of the design lifetime, with the following functional relationship:

$$M_{\text{propulsion}} = a + b \times M_{\text{propellant}}^{\frac{2}{3}} \tag{26}$$

where a and b are constants and depend on the particular design of the propulsion subsystem. They do not vary with the design lifetime (Fig. 12).

Attitude Determination and Control Subsystem

The attitude determination and control subsystem (ADCS) measures and controls the spacecraft's angular orientation. This subsystem stabilizes the spacecraft in desired orientations during different mission phases despite disturbance torques (thruster misalignment, aerodynamic torque, solar radiation torque, etc.) and is also used to reorient the spacecraft to point the payload in different directions (slew maneuvers). Its mass accounts for on average 6% of a satellite dry mass (see Table 1).

The issue of concern in this section is how the ADCS scales with the spacecraft design lifetime. The selection and sizing of the ADCS is driven by requirements on accuracy and range of angular motion both in terms of determination and control. For a three-axis stabilized spacecraft, the torque capability or control authority of reaction and momentum wheels is determined by the magnitude of the disturbance torques and the elements of the spacecraft inertia matrix:

$$T_{\text{ADCS}} = f(T_{\text{dist}}, \boldsymbol{I}, \ldots) \tag{27}$$

For a mass M with an orthogonal coordinate system (x, y, z) located at its center of mass, the moment of inertia about the z axis, for instance, is given by

$$I_{zz} = \int_M (x^2 + y^2) \times \mathrm{d}M \tag{28}$$

As the design lifetime increases, the spacecraft mass increases (EPS, thermal subsystem, propellant, etc.); hence, the elements of its inertia matrix increase. Consequently, the ADCS has to be redesigned for a larger torque capability. How can we relate the torque capability of a wheel to its mass? Answering this question would provide an insight into the relationship between the ADCS mass and the spacecraft design lifetime. This step unfortunately is not straightforward. In the absence of a physically based rationale for relating the ADCS mass to the design lifetime, we will use as a substitute the mass estimate relationship provided in Table 1 to evaluate the mass of a three-axis ADCS:

$$M_{\text{ADCS}} = 0.06 \times M_{\text{dry}} \tag{29}$$

Structures

The function of the spacecraft structure is to provide mechanical support to all subsystems within the framework of the spacecraft configuration. It also satisfies the subsystem requirements, such as alignment of sensors, actuators, antennas, etc., and the system requirements for launch vehicle interfaces and integration.[3] The spacecraft structure is a major contributor to the spacecraft dry mass and accounts for 21% of its dry mass (see Table 1). As in the preceding sections, we are interested in how the spacecraft structure mass scales with the design lifetime. To address this question, we start by examining the sources of structural requirements. Structures must endure mechanical loads in different environments, from manufacturing, to launch and normal operations.[10] The environments from which the structural requirements are derived are listed in Table 4.

None of these structural requirements can clearly relate the spacecraft design lifetime to the structural requirements and,

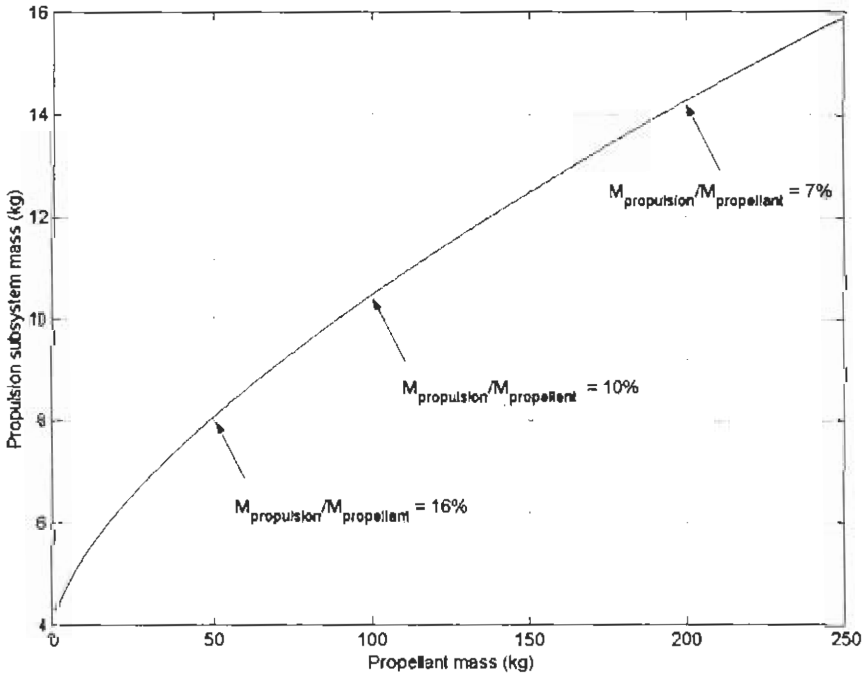

Fig. 12 Typical propulsion subsystem mass as a function of propellant mass: $a = 4$ and $b = 0.3$.

Table 4 Sources of structural requirements by mission phase (adapted from Ref. 1)

Environment/phase	Source of requirements
Manufacturing and assembly	Handling fixtures, stresses induced by welding, etc.
Transport and handling	Crane or dolly reactions, land, sea, air transport environments
Testing	Vibrations and acoustic tests, test fixtures reaction loads
Prelaunch	Handling during stacking sequence, preflight checks
Launch	Steady-state booster acceleration, acoustic noise, transient loads during booster ignition, burnout, pyrotechnic shock from separation events

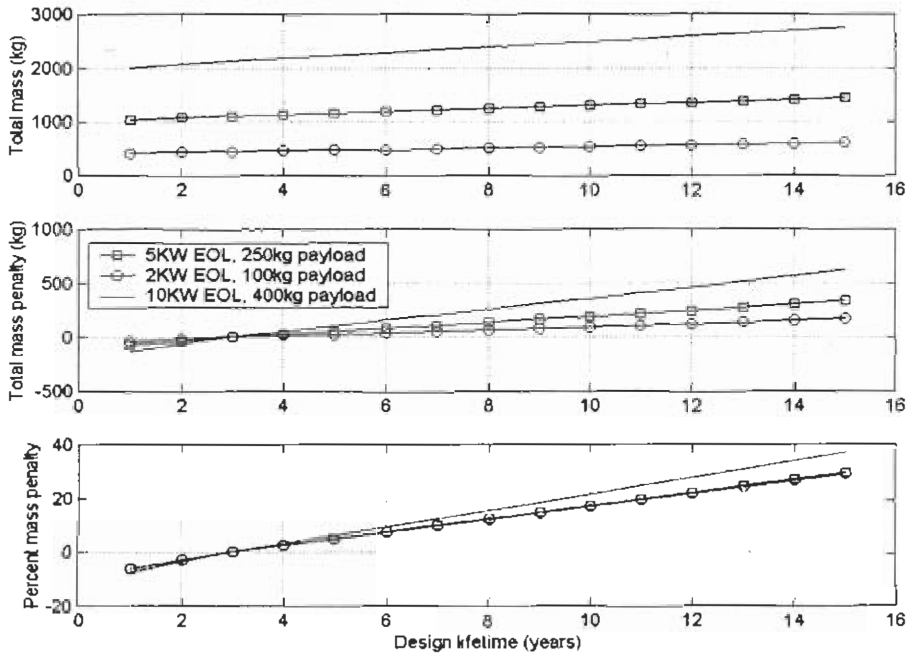

Fig. 13 Spacecraft total mass, mass penalty, and percent mass penalty as a function of the design lifetime: spacecraft in GEO, mission reliability 95%, three-axis stabilized, GaAs cells, Ni–H$_2$ batteries, reference mission three years.

consequently, to the spacecraft structure mass. It is reasonable to assume that the spacecraft structure scales with the design lifetime because the different subsystems enclosed within or supported by the structure, as well as the consumables, scale with the design lifetime (EPS, thermal, propulsion, propellant). It is not obvious, however, how the structure mass scales with the design lifetime. The least arbitrary approach is to maintain the mass estimate relationship given in Table 1 relating the spacecraft structure to the satellite dry mass

$$M_{\text{struct}} = 0.21 \times M_{\text{dry}} \tag{30}$$

Spacecraft Mass Profile

The spacecraft mass profile as a function of the design lifetime can now be illustrated by combining the effects of the design lifetime on the different subsystems as already discussed. The independent variables include orbit type and related parameters (eclipse duration, number of batteries charge/discharge cycles, degradation per year of solar arrays, ΔV_{yr}), solar cell type and battery type, power at EOL, mission reliability, type of attitude control, and payload mass.

The spacecraft dry mass and total mass (loaded mass) are calculated as follows:

$$M_{\text{dry}} = M_{\text{EPS}} + M_{\text{thermal}} + n \times M_{\text{TT\&C+CD\&H}} + M_{\text{ADCS}} + M_{\text{propulsion}} + M_{\text{struct}} + M_{\text{payload}}$$

$$M_{\text{tot}} = M_{\text{dry}} + M_{\text{propellant}} \tag{31}$$

Figure 13 shows typical spacecraft mass profile, mass penalty, and percent mass penalty as a function of the design lifetime. Note, for instance, that designing a spacecraft for 3 years instead of 15 years results in a mass saving of the order of 40%. Conversely, a mass penalty of 40% is incurred if a mission is initially designed for 15 years instead of 3 years. The next step is to translate this mass penalty, or mass saving, into a cost penalty, or cost saving. This is undertaken in the next section.

Cost to IOC and Cost per Operational Day

This section is based to a large extent on Ref. 11, as well as on Ref. 12.

In this section, we are interested in isolating the effect of the design lifetime on the spacecraft cost. We will proceed by translating the various mass profiles established earlier into spacecraft cost profiles as a function of the design lifetime. To do so, an understanding of the rationale, advantages, and limitations of cost estimate relationships, as well as the various components of a spacecraft cost, is required. The following paragraphs summarize the basics of cost modeling.

A spacecraft's cost depends on its size, complexity, technology readiness, design lifetime, schedule, as well as other characteristics. Space systems have specific costs (cost-per-unit weight) of the order of \$70,000/kg (Ref. 12). Specific costs, however, are not sufficient for predicting the real costs of spacecraft. Over the years, several governmental organizations have developed cost estimate relationships (CERs) that relate spacecraft cost or subsystem cost to physical, technical, and performance parameters. The CERs are based on an appropriate historical database of past satellite programs. The basic assumption of parametric cost modeling is that satellites will cost next time what they cost the last time. CERs include both nonrecurring and recurring costs associated with a space system. Nonrecurring costs are commonly referred to as the research, development, test, and evaluation (RDT&E) costs. These costs include the design, analysis, and test of prototypes and qualification units. Recurring costs include the cost to produce flight units. They are commonly referred to as the theoretical first unit (TFU) costs. This concept represents the cost of the first space-qualified satellite. Typical CERs include the range of the parameters used to develop the correlations between the subsystems characteristics and their cost, the CER itself, and the associated standard error (SE). An example is given in Table 5.

Launch costs, on the other hand, are derived from published look-up tables. Reference 13 is the guide for launch systems characteristics and costs. Another approach to modeling launch cost is to evaluate an average cost per kilogram to orbit. For instance, the average cost to LEO per kilogram for both United States and European launchers is approximately $10,000. Finally, the cost to IOC is given by

$$\text{cost}_{\text{IOC}} = \text{TFU} + \text{RDTE} + \text{cost}_{\text{launch}} \tag{32}$$

Using the linear extrapolation of the launch cost, the cost to IOC can be plotted as a function of the spacecraft design lifetime. Figure 14 shows the range of uncertainty in the cost estimation of the spacecraft recurring and nonrecurring costs.

Figure 15 shows typical spacecraft cost to IOC, cost penalty, and percent cost penalty as a function of the design lifetime. Note, for instance, that designing a spacecraft for 3 years instead of 15 years results in a cost saving of the order of 35%. Conversely, a cost penalty of 35% is incurred if a mission is initially designed for 15 years instead of 3 years. Figure 15 provides an answer to the question we set out to investigate in this paper, namely, how does the design lifetime requirement impact the total system (mass and) cost to IOC? The results confirm that the design lifetime is indeed a key driver of the space system cost and illustrate its particular impact on the various subsystems (EPS, thermal, propulsion, etc.). We can now define the cost per operational day of a spacecraft as follows:

$$\text{cost}_{/\text{day}} = \frac{\text{cost to IOC}}{\text{design lifetime (days)}} \tag{33}$$

(Fig. 16). This metric corresponds to uniformly amortizing the cost to IOC over the entire intended mission duration (without accounting for the time value of money).

Table 5 CER for estimating subsystems TFU and RDT&E costs (adapted from Ref. 1)

Component	Parameter x, kg	Range, kg	TFU,[a] \$1000	SE, %	RDTE,[a] \$1000	SE, %
Structure	Weight	[54, 560]	13.1x	36	157$x^{0.83}$	38
EPS	Weight	[31, 573]	112$x^{0.763}$	44	62.7x	57

[a]Fiscal year 2000.

Within the interval of the design lifetime considered, the cost per operational day decreases monotonically. In the absence of other metrics, the cost per operational day justifies pushing the boundary of the design lifetime and designing spacecraft with increasingly longer lifetimes. It also suggests that a customer is always better off requesting the contractor to provide the maximum design lifetime

$$T_{\text{life}} = T_{\text{life-max}} \tag{34}$$

This, however, is not necessarily true. Launching spacecraft with increasingly longer design lifetimes raises the risk for the satellite of becoming technically and commercially obsolete before the end of its mission. Thus, in specifying the design lifetime requirement, decision makers have to assess this risk of loss of value due to both the obsolescence of their product's technology base as well as the likelihood of changing market needs, or the volatility of the market the system is serving, during the system's operational lifetime. These issues will be explored in a subsequent paper.

Limitations

The preceding analysis presents several limitations that degrade the accuracy of the results. First, to isolate and capture the effects of the design lifetime on the spacecraft mass and cost, a limited number of parameters were considered in the analysis, instead of the plethora of variables that subsystems experts typically have to trade and optimize. This was done to maintain a manageable size analysis and to avoid drowning the key parameters and effects in background clutter.

The second limitation results from the use of mass estimate relationships, such as in the case of the spacecraft structure. Although it is clear that the spacecraft structure, for instance, scales with the design lifetime by the fact the different subsystems enclosed within or supported by the structure scale with the design lifetime (EPS, thermal, propellant, etc.), it is not possible to relate the spacecraft structure's mass to the design lifetime without taking into account particular details about the mission or the spacecraft configuration and layout. In other words, a preliminary design of the spacecraft is required to estimate reasonably the mass of the spacecraft structure. In light of the objectives set forth in the Introduction and summarized earlier, such an analysis is beyond the scope of this study. In the absence of quantifiable physical arguments for relating a subsystem's mass to the design lifetime, mass estimate relationships were used as the least arbitrary way to proceed with the analysis.

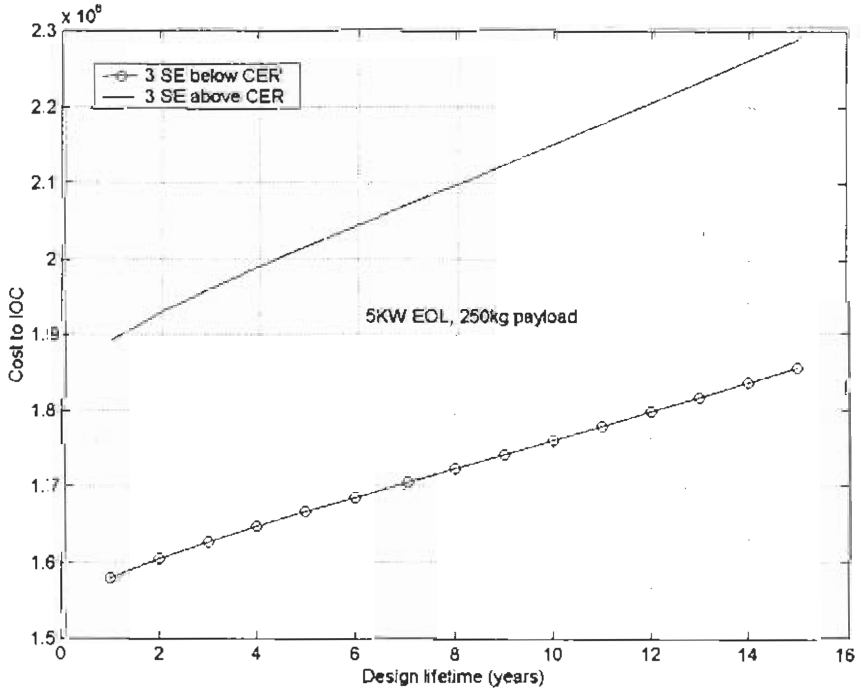

Fig. 14 Cost to IOC for a LEO spacecraft as a function of the design lifetime (three SE above and below the nominal CER output, same parameters as in Fig. 13).

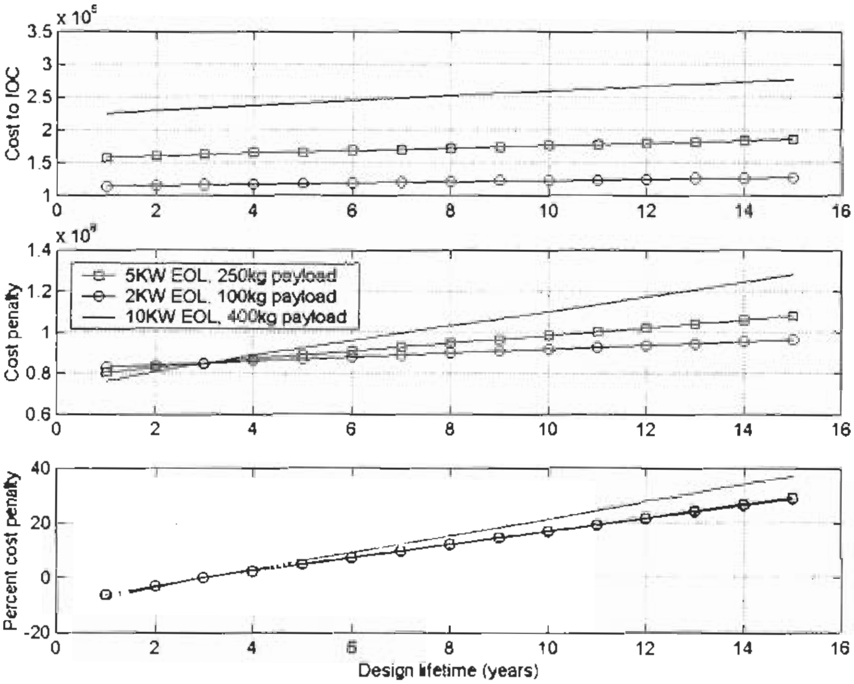

Fig. 15 Cost to IOC, cost penalty, and percent cost penalty as a function of the design lifetime (same parameters as in Fig. 13).

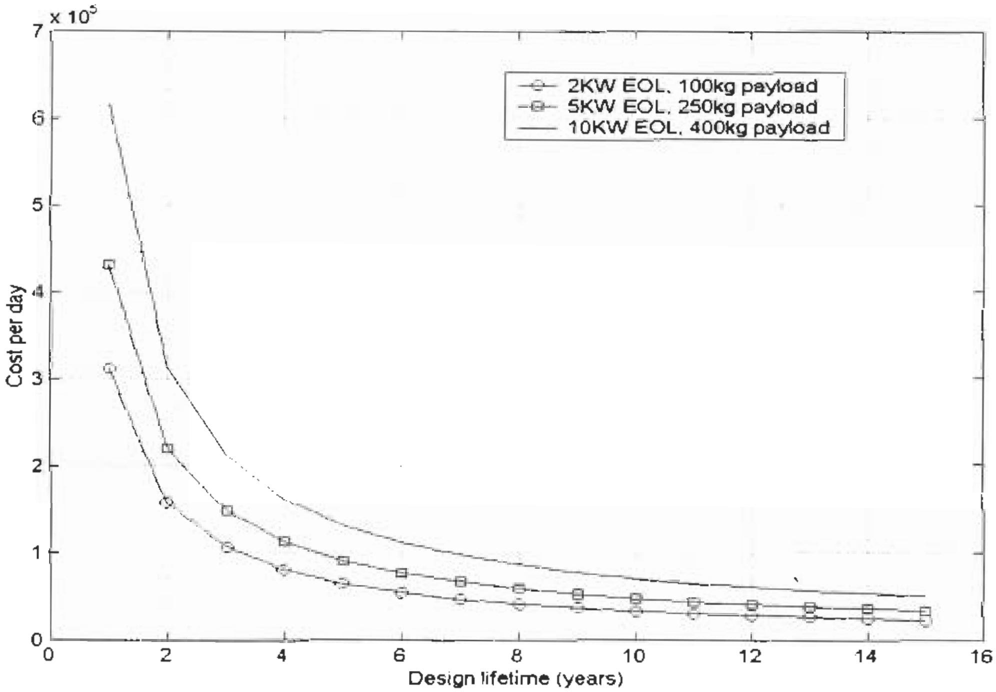

Fig. 16 Cost per operational day as a function of the design lifetime (same parameters as in Fig. 13).

The third limitation is due in part to the use of cost estimate relationships and dollars per pound to estimate launch costs. This resulted in smooth or continuous cost profiles instead of the discontinuous profiles that would be obtained in reality because of the performance and cost of existing launch systems, for example, \$13 million for less than 1000 lb to LEO on Pegasus XL and \$22 million for less than 3000 lb to LEO on Taurus. The availability and use of commercial-off-the-shelf hardware, which exists in discrete performance bins and does not necessarily match the customer's needs exactly, will also render discontinuous both the mass and cost profile of a spacecraft as a function of the design lifetime.

Some of the limitations discussed render the task of building generic models relating the spacecraft mass and cost to the design lifetime very challenging. However, in practice, the aforementioned inaccuracies will be attenuated when, during the conceptual design phases of a particular spacecraft, designers evaluate the mass and cost of their particular design at discrete values of the design lifetime, for example, three, five, seven, nine years, all else being equal (performance and reliability). Thus, more accurate estimates could be obtained for the mass and cost of the spacecraft, or its cost per operational day and help guide the selection of the design lifetime.

Conclusions

This paper explored the impacts of the design lifetime on the spacecraft mass and cost to IOC. It first examined how different subsystems scale with the design lifetime, using physically based arguments whenever possible and mass estimate relationships in other instances. The data were then transformed to generate spacecraft mass and cost profiles as a function of the design lifetime. Preliminary results confirm that the design lifetime is a key requirement in sizing various subsystems. For instance, a mass and cost penalty of 30–40% is typically incurred when designing a spacecraft for 15 years instead of 3 years, all else being equal. It was also shown that the cost per operational day decreases monotonically with the design lifetime. This finding justifies pushing the boundary of the design lifetime and designing spacecraft with increasingly longer lifetimes. It also suggests that a customer is always better off asking the contractor to provide the maximum design lifetime achievable. This, however, may not always be the case. The decision regarding the design lifetime requirement should incorporate external factors such as the obsolescence of the technology embedded in the spacecraft, the relationship between technology obsolescence and market share, and the volatility of the market the mission is serving in the case of a commercial satellite.

Acknowledgments

This work was supported in part by the Defense Advanced Research Projects Agency. The authors would like to thank Peter Young (retired), Daniel Vilani, and Michael Smith from Boeing Satellite Systems, as well as Elisabeth Lamassoure, Karen Marais, and Chris Carr for their valuable comments. A considerable intellectual debt is due to James Wertz and Wiley Larson for the material used and for their discussion of the design lifetime requirement. This work is a continuation of their discussion.

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